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RESEARCH MEMORANDUM

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To
By Authority of NASA TD 70-688 Date 11/25/70

CALCULATED PERFORMANCE OF A DIRECT-AIR NUCLEAR TURBOJET-POWERED AIRPLANE USING A SPLIT-FLOW REACTOR AND A SEPARATED-TYPE SHIELD

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NATIONAL ADVISORY COMMITTEE
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By R. B. Doyle

SUMMARY

An analysis was made to estimate the performance of a direct-air nuclear turbojet-powered airplane using a split-flow reactor and a separated-type shield. The analysis was for flight Mach numbers of 0.9 and 1.5 and covered a range of altitudes, reactor-wall temperatures, turbine-inlet temperatures, compressor pressure ratios, and airplane lift-drag ratios.

For a flight Mach number of 0.9, sea-level altitude, a reactor-wall temperature of 2000° R, a turbine-inlet temperature of 1800° R, and an airplane lift-drag ratio of 7, the calculations indicated that an airplane having a gross weight of 342,000 pounds would be required to carry a pay load of 20,000 pounds. In order to carry the same pay load at a flight Mach number of 1.5, an altitude of 30,000 feet, a reactor-wall temperature of 2300° R, a turbine-inlet temperature of 2100° R, and an airplane lift-drag ratio of 5, an airplane having a gross weight of 436,000 pounds would be required.

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INTRODUCTION

Analyses are being made at the NACA Lewis laboratory of various types of propulsion system utilizing a nuclear reactor as the energy source. One system that is being studied is the direct-air turbojet cycle for which some results were presented at a flight Mach number of 0.9 in reference 1. In reference 2, a comparison of three cycles was made, and in this study also the results for the direct-air turbojet cycle are presented only for flight at a Mach number of 0.9.

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The radiation shields that were considered in references 1 and 2 were, in general, the wrap-around or integral-type shield and resulted in very heavy shield weights for the large diameters that are frequently required for an air-cooled reactor. Some check calculations indicated that, with these shields and the reactor-wall temperatures considered (up to 2500° R), flight at a Mach number of 1.5 was quite impractical; the airplane gross weights required to carry a pay load of 20,000 pounds were of the order of 1,000,000 pounds.

More recent shielding theories (references 3 and 4) indicate that the shield weight can be greatly reduced from the values used in references 1 and 2 by using a separated-type shield wherein part of the radiation shielding is placed around the reactor and part around the airplane-crew compartment. In addition, for the same reactor diameter and hence for the same shield weight, larger reactor air-handling capacities can be realized by using a split-flow-type reactor.

Because of the sizeable reduction in shield weight that now appears possible by use of the separated-type shield and split-flow-reactor arrangement, additional calculations have been made for the direct-air turbojet cycle and the results are presented herein. The calculations now include results for a flight Mach number of 1.5 in addition to results for a flight Mach number of 0.9 and cover a range of altitudes, reactor-wall temperatures, turbine-inlet temperatures, compressor pressure ratios, and airplane lift-drag ratios.

Airplane lift-drag ratio is included as a primary variable in this analysis because it is felt that at the present time there is insufficient information available, especially in the supersonic-speed range, to make a realistic assumption of single values of airplane lift-drag ratio for a given flight condition.

ANALYSIS

Description of Power Plant

A schematic diagram of the turbojet engine is shown in figure 1. Air enters the engine through an inlet diffuser and passes through the compressor into the reactor where it is heated by contact with the walls of the reactor flow passages. From the reactor, the air expands through the turbine and the exhaust nozzle as in the conventional turbojet engine. Inasmuch as the optimum performance of the system occurs at relatively high compressor pressure ratios, an intercooler was included between compressor stages.

Assumptions

2046 Engine and airplane. - Some of the pertinent assumptions that were made for the engine and the airplane are listed in the following table:

Diffuser recovery factor (ratio of actual to theoretical total pressure)	0.96
Compressor small-stage efficiency	.88
Turbine adiabatic efficiency	.90
Exhaust-nozzle velocity coefficient	.96
Intercooler cooling effectiveness	.50
Ratio of airplane structure weight to gross weight	.35
Disposable load, lb	20,000
Lift-drag ratio of complete airplane	Variable

The method of evaluating engine weight was the same as that used in reference 1. The engine weights, as calculated by this method, are representative of the lightest of current turbojet engines.

The airplane lift-drag ratio was varied for most calculations over a range, which, it was felt, included practical design values based on wing loading and either landing or take-off limitations.

Reactor and shield. - The reactor and the reactor-shield configuration that were considered in this analysis are schematically shown in figure 2. The selection of shielding materials and their densities and thicknesses was not based on any nuclear calculations made at the NACA, but rather on the results of other investigators (references 3 and 4). The choice of a specific geometrical arrangement was made simply for the purpose of determining a total shield weight. The reactor is a cylindrical split-flow type and was assumed to have a length-to-diameter ratio of 0.9 and a free-flow area ratio of 0.5. In the split-flow arrangement, the reactor is cut by a transverse gap midway between the ends. The coolant flows into this gap, through the reactor, and out both ends. The reactor core is enclosed around the circumference and the ends by a 3-inch-thick reflector.

The separated shield considered herein consists essentially of a jacket of relatively low mean-density material surrounding the reflector and the reactor and a separate crew shield. The reactor and the reflector were surrounded by 4 inches of lead, which in turn was surrounded by 4 feet of material having a specific gravity of 0.85. Some reactor-shield weight saving was accomplished by rounding the corners of the shield with a 4-foot radius.

The reactor shield as schematically shown in figure 2 does not have provision for ducts for passing the air in and out of the reactor. In calculating the reactor-shield weight, however, the unducted reactor-shield weights were increased by 15 percent to allow for the air ducting.

The crew compartment was considered to be a hollow lead cylinder closed on the end facing the reactor and weighing 50,000 pounds.

Methods

The performance of the nuclear turbojet-powered airplane was evaluated on the basis of the minimum airplane gross weight required to carry a disposable load or pay load of 20,000 pounds.

For each combination of flight Mach number, altitude, reactor-wall temperature, and turbine-inlet temperature, the compressor pressure ratio and reactor pressure drop (a function of reactor-inlet Mach number) were varied to determine the minimum airplane gross weight and the corresponding optimum engine operating conditions.

For all calculations, the turbine-inlet temperature was assumed to be 200° R below the reactor-wall temperature. Although no systematic study was made to determine the effect of this temperature difference (reactor-wall minus turbine-inlet) on the system performance, a few calculations on the direct-air turbojet indicate that 200° R is about optimum.

RESULTS AND DISCUSSION

Effect of lift-drag ratio. - The effect of airplane lift-drag ratio on airplane gross weight at various altitudes and for flight Mach numbers of 0.9 and 1.5 is illustrated in figure 3. The disposable load or pay load was assumed to be 20,000 pounds. Reactor-wall and turbine-inlet temperatures were selected for each flight condition so that the resulting airplane gross weights were always either less than or only slightly over 1,000,000 pounds. Thus for the less severe flight condition (Mach number 0.9, fig. 3(a)), the reactor-wall temperature is 2000° R and the turbine-inlet temperature is 1800° R; for the more severe flight condition (Mach number 1.5, fig. 3(b)), the reactor-wall temperature is 2300° R and the turbine-inlet temperature is 2100° R.

The reactor-core diameter is a dependent variable in all calculations and is determined by the heat-transfer requirements of the cycle. Dashed lines of constant reactor diameter are included on this figure.

At all altitudes and flight Mach numbers, the airplane gross weight and the reactor size increase rapidly with decreasing lift-drag 2046 ratio. At sea level and a flight Mach number of 0.9, the airplane gross weight to carry a 20,000 pay load would be less than 500,000 pounds for a lift-drag ratio as low as 5.5. At the same flight Mach number and at altitudes of 30,000 and 50,000 feet, the airplane gross weight would be slightly less than 500,000 pounds for lift-drag ratios of 7 and 15, respectively.

At a flight Mach number of 1.5 and at the higher reactor-wall temperature (2300° R), airplane lift-drag ratios of 4.5 and 9 at altitudes of 30,000 and 50,000 feet, respectively, would result in airplane gross weights of about 500,000 pounds.

The reactor diameters from figure 3 for airplane gross weights of 500,000 pounds would be about 4.5 and 5.2 feet for flight Mach numbers of 0.9 and 1.5, respectively. All curves in figure 3 were discontinued at a reactor diameter of 3 feet, which was the minimum size considered in this analysis.

Additional information on probable lift-drag ratios is required before the flight condition can be determined that results in minimum airplane gross weight. If the design lift-drag ratio is to be based on landing or wing-loading limitations, the maximum possible lift-drag ratios for a given flight Mach number will vary with altitude. For constant wing loading or constant landing speed, the design lift-drag ratio tends to increase with increase in design altitude at constant flight Mach number. Thus to determine the flight altitudes at which minimum airplane gross weight occurs, it is necessary to be able to predict the design lift-drag ratios for various altitudes. If, for example, the maximum airplane design lift-drag ratio obtainable at a flight Mach number of 1.5 and an altitude of 50,000 feet is 9, the airplane gross weight according to figure 3(b) would be 500,000 pounds. If, however, the maximum lift-drag ratio obtainable at the same flight Mach number but at an altitude of 30,000 feet is considerably lower, for example 6, the airplane gross weight would be only 350,000 pounds despite the lower airplane lift-drag ratio.

It might be emphasized here that the airplane gross weights shown herein that are required to carry a 20,000 pound pay load are considerably lower than indicated in references 1 and 2, and that this difference is due almost entirely to the use in this analysis of the low shield weights associated with the split-flow reactor and separated-shield arrangement.

Effect of reactor-wall temperature. - The effect of reactor-wall temperature on airplane gross weight for altitudes of sea level and

50,000 feet at a flight Mach number of 0.9 is shown in figure 4. An airplane lift-drag ratio of 7 was chosen arbitrarilly for the sea-level calculations and a lift-drag ratio of 10 was likewise chosen for the calculations at 50,000 feet, inasmuch as the design lift-drag ratio for constant wing loading or constant landing speed would be expected to increase by some amount with altitude, as previously mentioned. The turbine-inlet temperature in each aase is 200° R below the reactor-wall temperature.

For the sea-level altitude case, the airplane gross weight and reactor size increase rather slowly with decreasing reactor-wall temperature between 2100° and 1900° R. Below about 1900° R, however, the airplane gross weight and the reactor size increase rapidly with decreasing temperature. The airplane weight is slightly below 500,000 pounds at sea-level altitude and a reactor-wall temperature of 1850° R. At an altitude of 50,000 feet, the airplane gross weight increases very rapidly with decreasing reactor-wall temperature and the airplane gross weight is over 1,000,000 pounds for reactor-wall temperatures below about 2090° R.

The effect of reactor-wall temperature on airplane gross weight for a flight Mach number of 1.5 and an altitude of 30,000 feet is shown in figure 5. The lift-drag ratio of the airplane is 5.0. The airplane gross weight varies from slightly over 600,000 pounds at a reactor-wall temperature of 2100° R to 360,000 pounds at a reactor-wall temperature of 2500° R.

The following table presents the airplane gross weights and corresponding reactor diameters for a few representative operating conditions as obtained from figures 3 to 5; also presented in the table are the reactor heat-release rates per unit volume, some engine weight and performance figures, and an airplane gross weight breakdown.

Flight Mach number	0.9	0.9	0.9	1.5	1.5
Altitude, (ft)	0	0	50,000	30,000	50,000
Airplane lift-drag ratio	7	7	15	5	10
Reactor-wall temperature[a], (^{o}R)	2000	1800	2000	2300	2300
Reactor core diameter, (ft)	3.6	6.2	4.5	4.6	4.6
Reactor heat release (kw/in.^{3})	6.8	2.1	1.7	5.5	2.2
Engine thrust, (lb)	48,000	83,200	31,800	81,400	42,500
Engine frontal area[b], (sq ft)	49	114	179	100	113
Reactor weight, (lb)	2000	12,000	5000	5000	5000
Reactor shield weight, (lb)	102,000	189,000	128,000	131,000	131,000
Crew shield weight, (lb)	50,000	50,000	50,000	50,000	50,000
Engine weight[c], (lb)	48,000	107,000	107,000	77,000	70,000
Airplane structure weight, (lb)	120,000	204,000	167,000	153,000	149,000
Pay load (lb)	20,000	20,000	20,000	20,000	20,000
Airplane gross weight, (lb)	342,000	582,000	477,000	436,000	425,000

[a]Turbine-inlet temperature 200^{o} R below the reactor-wall temperature.
[b]Based on compressor frontal area, no allowance made for nacelle.
[c]Compressor, turbine, intercooler, and shaft.

Optimum compressor pressure ratio. - In reference 1, it is shown that the optimum compressor pressure ratios for the nuclear-powered turbojet engine were about 40:1 for most conditions investigated. It is pointed out, however, and illustrated by a figure in reference 1 that the optimum compressor pressure ratio was largely a function of the shield weight and the turbine-inlet temperature.

In the present analysis, the compressor pressure ratio for optimum performance was found to be about 15:1 for most cases and below 25:1 in every case. This difference was due largely to the assumption of a lighter shield than was used in references 1 and 2. The following table lists the optimum compressor pressure ratios for some of the conditions that were investigated in the present analysis:

Flight Mach number	Altitude (ft)	Airplane lift-drag ratio	Turbine-inlet temperature ($^{\circ}R$)	Optimum compressor pressure ratio
0.9	0	7	1500	7
.9	0	7	1800	10
.9	30,000	7	1800	17
.9	30,000	10	1800	20
1.5	30,000	5	2100	14
1.5	50,000	10	2100	16

As previously mentioned, the design airplane lift-drag ratio for a given flight speed based on maximum wing loading or maximum landing or take-off speed would be a function of altitude; however, in the preceding table, for purposes of comparison of the optimum compressor pressure ratios, the same airplane lift-drag ratio is used in the first and third lines although these lines are for different altitudes.

The preceding table indicates that, for the range of conditions investigated and for the shield weights considered, the optimum compressor pressure ratio increases with increasing altitude, airplane lift-drag ratio, and turbine-inlet temperature.

SUMMARY OF RESULTS

The results of calculations on the performance of a direct-air nuclear turbojet-powered airplane using a split-flow reactor and a separated-type shield may be summarized as follows:

1. The airplane gross weight and the reactor size required to carry a specified pay load increased rapidly with decreasing reactor-wall temperature and airplane lift-drag ratio.

2. The following table gives, for some representative conditions investigated, the airplane gross weights and corresponding reactor diameter that are required to carry a disposable load of 20,000 pounds along with the unit volume reactor heat release rates, some engine performance and weight figures, and an airplane gross-weight breakdown.

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Flight Mach number	0.9	0.9	0.9	1.5	1.5
Altitude, (ft)	0	0	50,000	30,000	50,000
Airplane lift-drag ratio	7	7	15	5	10
Reactor-wall temperature[a], ($^{\circ}R$)	2000	1800	2000	2300	2300
Reactor core diameter, (ft)	3.6	6.2	4.5	4.6	4.6
Reactor heat release, (kw/in.^{3})	6.8	2.1	1.7	5.5	2.2
Engine thrust, (lb)	48,800	83,200	31,800	81,400	42,500
Engine frontal area[b], (sq ft)	49	114	179	100	113
Reactor weight, (lb)	2000	12,000	5000	5000	5000
Reactor shield weight, (lb)	102,000	189,000	128,000	131,000	131,000
Crew shield weight, (lb)	50,000	50,000	50,000	50,000	50,000
Engine weight[c], (lb)	48,000	107,000	107,000	77,000	70,000
Airplane structure weight, (lb)	20,000	204,000	167,000	153,000	149,000
Pay load, (lb)	20,000	20,000	20,000	20,000	20,000
Airplane gross weight, (lb)	342,000	582,000	477,000	436,000	425,000

[a]Turbine-inlet temperature 200° R below the reactor-wall temperature.
[b]Based on compressor frontal area, no allowance made for nacelle.
[c]Compressor, turbine, intercooler, and shaft.

3. The compressor pressure ratio for optimum performance of the system was about 15:1 for most conditions and not more than 25:1 for any of the conditions investigated.

Lewis Flight Propulsion Laboratory,
National Advisory Committee for Aeronautics,
Cleveland, Ohio, October 4, 1950.

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4. Aircraft Reactor Branch, Division of Reactor Development, U. S. Atomic Energy Commission: Application of a Water Cooled and Moderated Reactor to Aircraft Propulsion. August 18, 1950.

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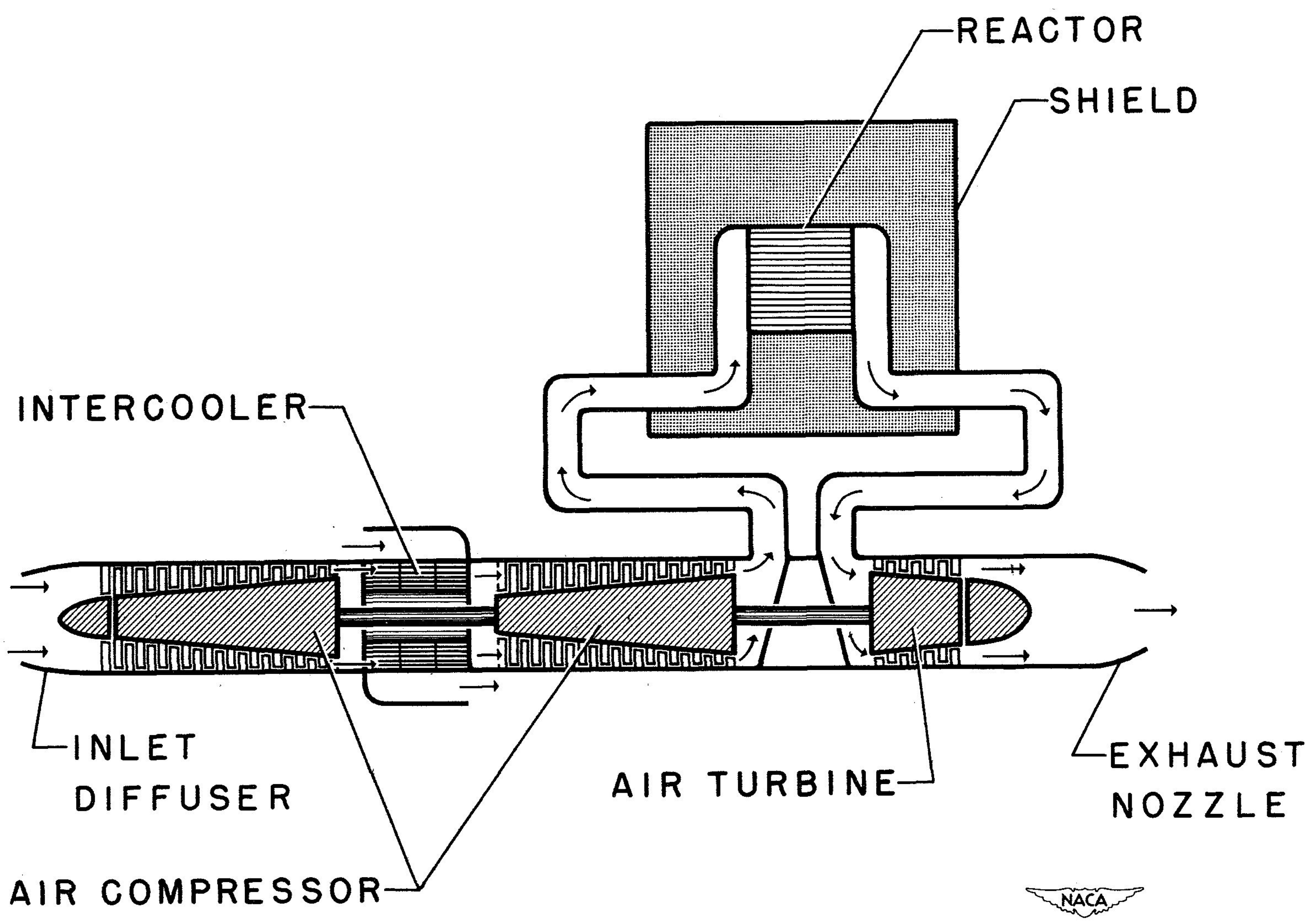

Figure 1. - Schematic diagram of direct-air cycle.

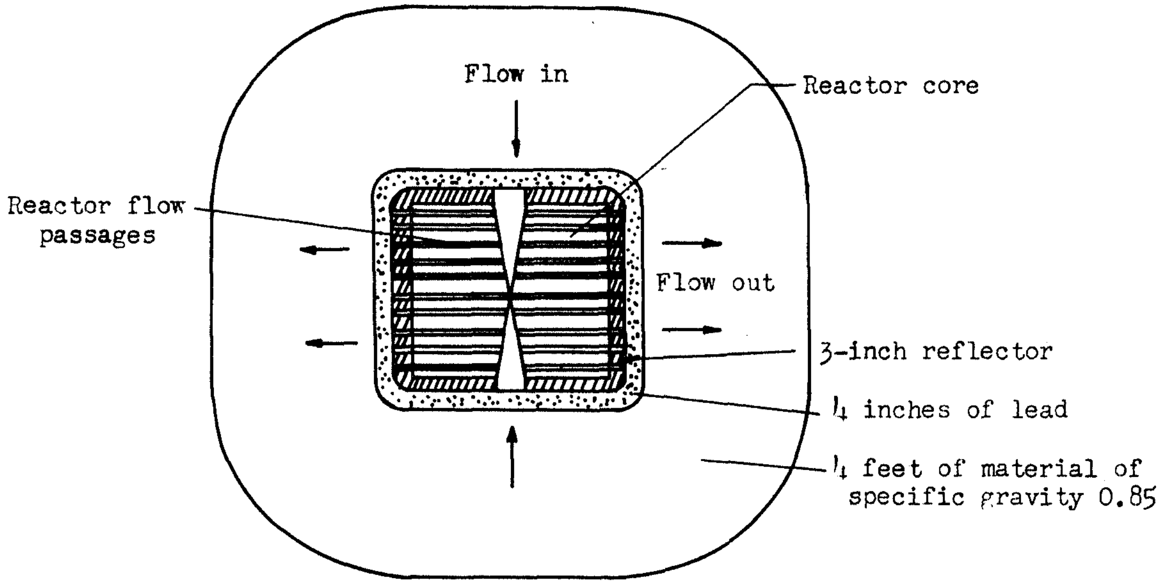

Figure 2. - Schematic diagram of reactor and reactor shield.

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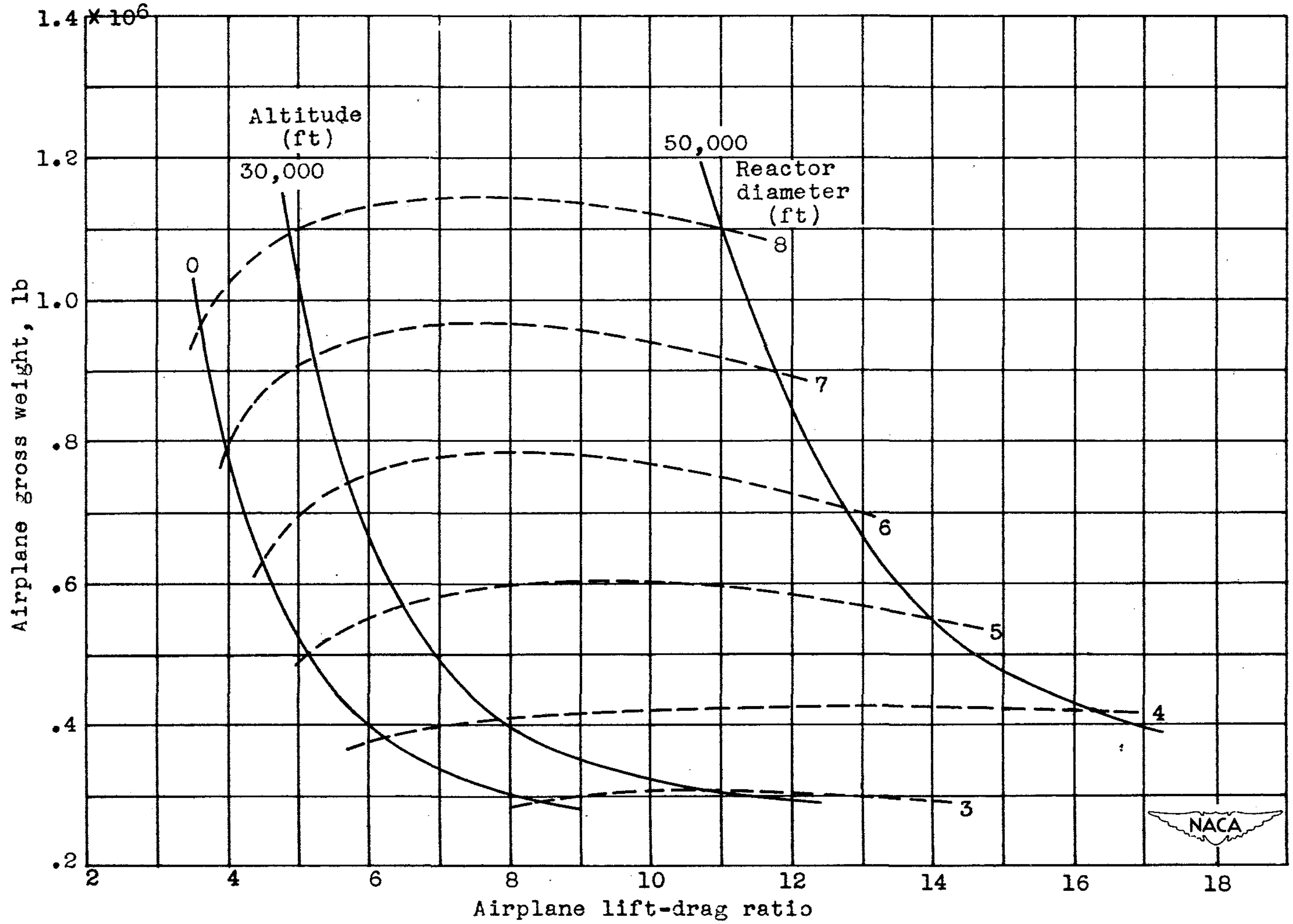
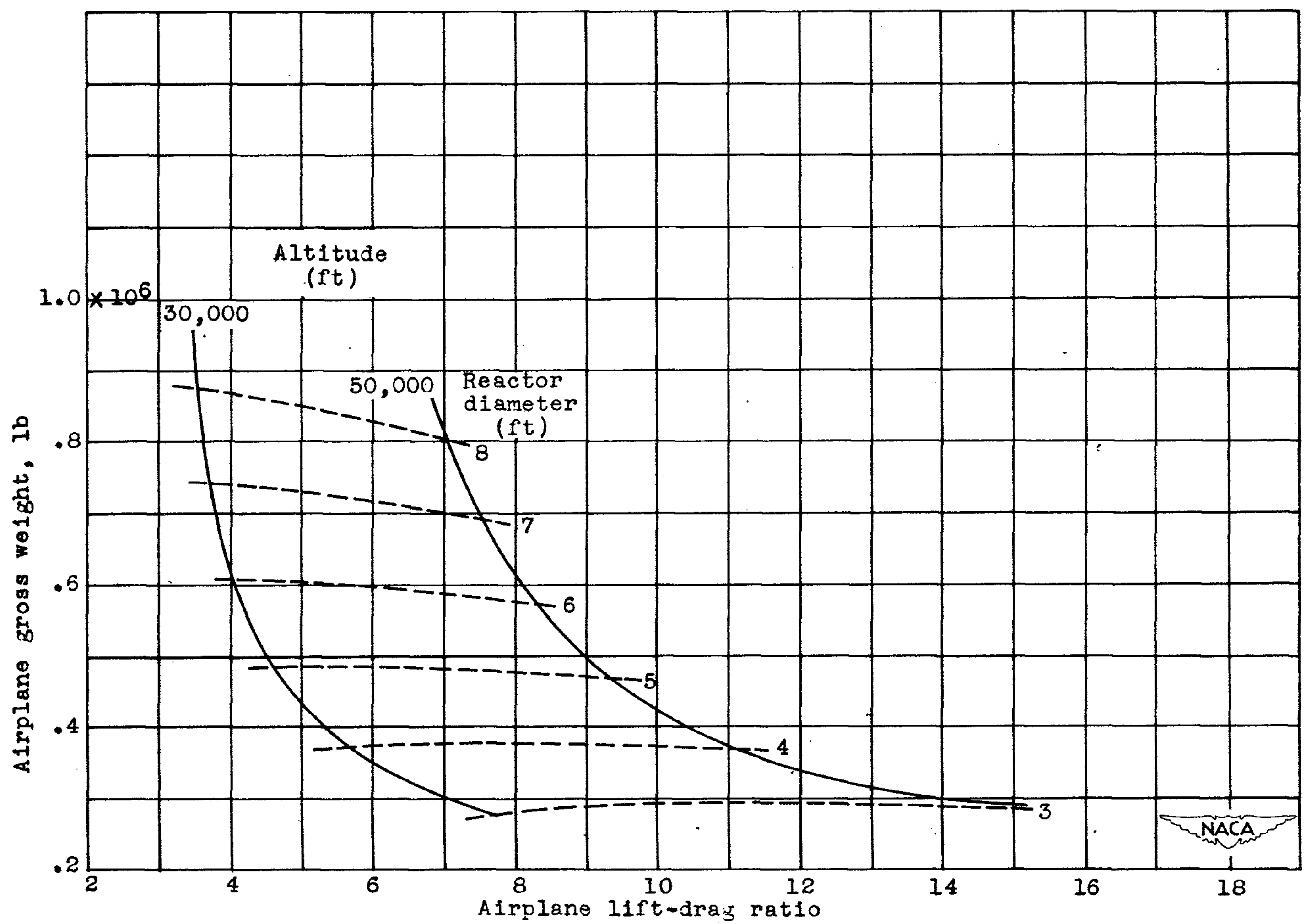


(a) Flight Mach number, 0.9; reactor-wall temperature, 2000° R; turbine-inlet temperature, 1800° R.

Figure 3. - Effect of airplane lift-drag ratio on airplane gross weight at various altitudes. Disposable load, 20,000 pounds.



(b) Flight Mach number, 1.5; reactor-wall temperature, 2300° R; turbine-inlet temperature, 2100° R.

Figure 3. - Concluded. Effect of airplane lift-drag ratio on airplane gross weight at various altitudes. Disposable load, 20,000 pounds.

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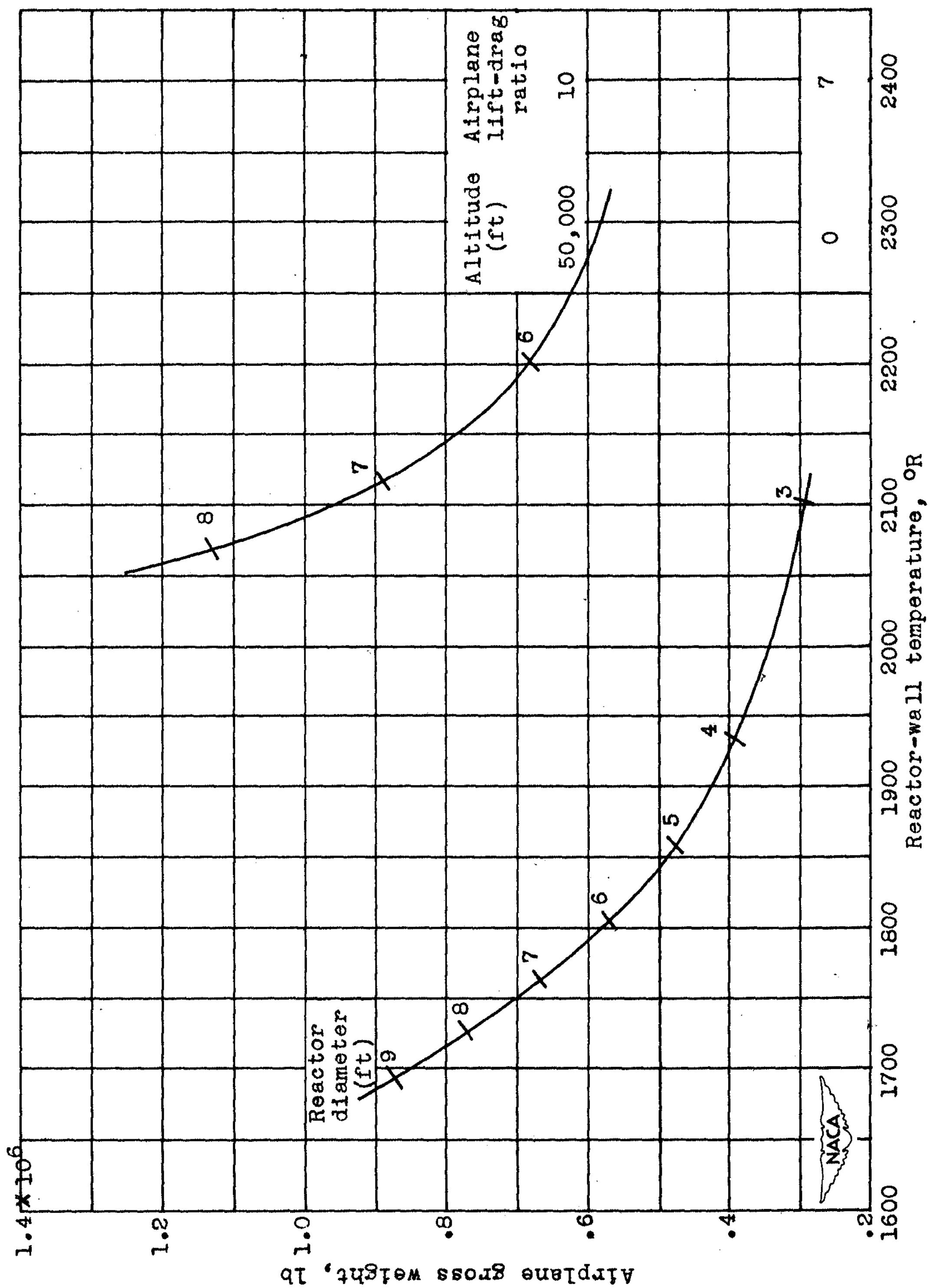

Figure 4. - Variation of airplane gross weight with reactor-wall temperature for various altitudes. Flight Mach number, 0.9; reactor-wall temperature minus turbine-inlet temperature, 200° R; disposable load, 20,000 pounds.

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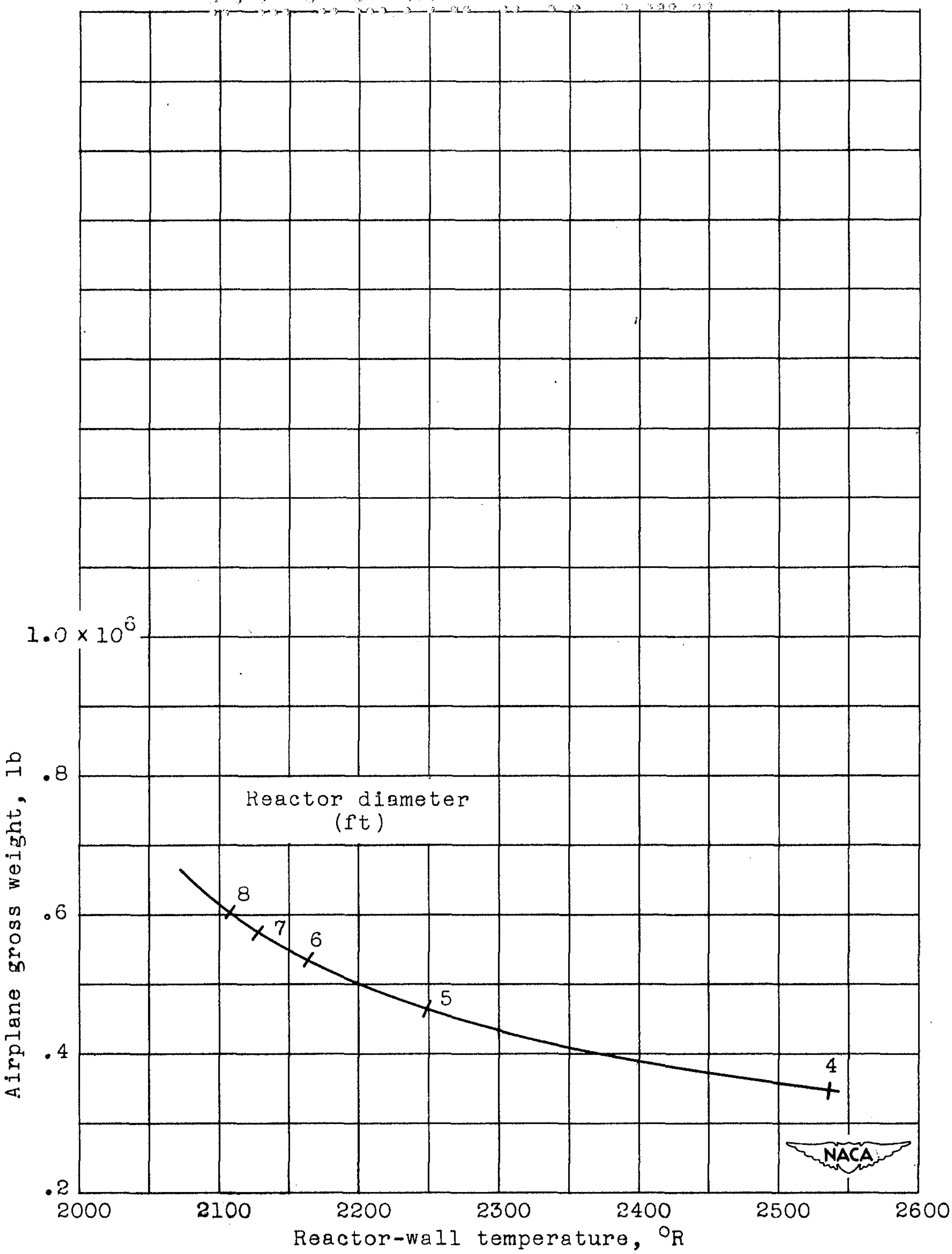

Figure 5. - Variation of airplane gross weight with reactor wall temperature. Flight Mach number, 1.5; altitude, 30,000 feet; airplane lift-drag ratio, 5.0; reactor-wall temperature minus turbine-inlet temperature, 200° R; disposable load, 20,000 pounds.

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